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MEMORANDUM

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By Robert G. Dorsch, John S. Serafini, Edward A. Fletcher and I. Irving Pinkel

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON
March 1959

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MEMORANDUM 1-11-59E

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SUMMARY

Pressure distributions associated with stable combustion of aluminum borohydride in the airstream adjacent to the lower surface of a 13-inch chord, two-dimensional, blunt-base wing were determined experimentally. The measurements were made with the wing at 2° angle of attack in a 1- by 1-foot tunnel at Mach numbers of 2.47 and 2.96.

Static-pressure increases along the lower surface and base caused by the combustion are presented along with the resultant lift increases. The lift-drag ratio of the wing was nearly doubled by the addition of heat.

The experimental values of lift during heat addition agree with those predicted by analytical calculations.

INTRODUCTION

An exploratory research program to determine the aerodynamic effects of the addition of heat to a supersonic stream has been conducted. The initial phases of the investigation were primarily analytical and were reported in references 1 and 2. The analytical calculations summarized in reference 2 indicate that practical benefits may be achieved by utilizing the increased static pressure associated with the addition of heat to the supersonic stream adjacent to an aerodynamic body. For example, it is shown that the lift-drag ratio of a wing in supersonic flight should be significantly increased by the direct addition of heat to the stream below the wing.

The next phase of the study was primarily concerned with finding a method of adding heat directly to the stream. In the experimental research reported in references 3 and 4 it was found that aluminum borohydride injected into the high velocity stream adjacent to the top wall of a supersonic wind tunnel would burn stably upon ignition. No flameholders, other than the tunnel wall and its associated boundary layer, were necessary to stabilize the combustion.

In the present phase of the investigation the techniques of references 3 and 4 are being employed to obtain external combustion adjacent to several aerodynamic bodies in a small (1- by 1-ft) supersonic wind tunnel in order to study the aerodynamic effects of heat addition. The aerodynamic effects produced by the combustion of aluminum borohydride in a Mach 2.46 airstream adjacent to a 13-inch and a 25-inch flat-plate model are summarized in reference 5. The results obtained with a body of revolution at Mach 2.47 are given in reference 6.

The results obtained with a third model, a two-dimensional, 6-percent thick, blunt-base supersonic wing are presented in this report. Reference 7 showed that the low-drag airfoil section (306-T) chosen had good aerodynamic characteristics at the Mach numbers employed in this investigation. The results will therefore indicate the lift-drag-ratio (L/D) improvement possible when heat is added below a conventional wing which initially has good aerodynamic qualities. As in the previous experimental studies, the heat addition was accomplished by burning aluminum borohydride in the adjacent supersonic stream. The location of the heat-addition region below the wing was arbitrarily chosen to provide primarily a lift increase rather than a drag reduction in order to facilitate comparison with the theoretical calculations of reference 2.

This report summarizes the static-pressure changes at the wing surface caused by external combustion of aluminum borohydride. The measurements were made in one of the Lewis 1- by 1-foot supersonic tunnels with the wing at an arbitrary 2° angle of attack and with free-stream Mach numbers of 2.47 and 2.96. The lift and drag of the wing during combustion are computed from the pressure data and are compared with the nonburning values. In addition, the measured lift coefficients are compared with those calculated analytically.

APPARATUS AND PROCEDURE

The apparatus and procedures employed in this investigation are similar in many details to those described in reference 5. As in reference 5, a 1- by 1-foot nonreturn-type supersonic tunnel was employed. Aluminum borohydride was burned for 2- to 3-second periods below the lower surface of a two-dimensional wing. The tests were conducted at free-stream Mach numbers of 2.47 and 2.96.

Model and Instrumentation

The two-dimensional wing model was installed in the test section at a 2° angle of attack in the vertical plane as shown in figure 1. In order to facilitate direct and schlieren photography, the model was mounted across the schlieren window of the tunnel by attaching it to the window retaining rings. The coordinates of the airfoil section (approximately circular arc with blunted trailing edge) are given in table I. Other pertinent dimensions of the model are:

Maximum thickness to chord ratio	0.06
Chord length, in.	13
Span, in.	11.94
Maximum thickness (at 67-percent chord), in.	0.78
Blunt trailing-edge thickness, in.	0.46
Leading-edge angle (total included)	$6^{\circ}59'$
Distance from leading edge to fuel orifices, in.	3.5
Distance from central to outboard fuel orifices, in.	2.0

The arrangement of static-pressure taps on the model surface is shown in figure 2. Rather large static orifice diameters (0.048 in.) were used to reduce the possibility of plugging by ash deposits on the model surface resulting from the external combustion.

The static pressures in the absence of combustion were measured with dibutylphthalate differential manometer boards. Changes in wing surface and base static pressures caused by combustion were measured with strain-gage-type and NACA standard-base six-capsule manometer differential pressure transducers.

Fuel Injection and Ignition

The method of injecting fuel into the airstream adjacent to the wing was basically similar to that described for the flat-plate model of reference 5. The fuel injector (fig. 1) was loaded with 25 to 35 cubic centimeters of aluminum borohydride prior to each run and pressurized with helium. The liquid was injected under pressure (100 lb/sq in. gage) into the airstream through three fuel orifices 0.016 inch in diameter located in the lower surface of the model as shown in figure 2. The injection period was of 2- to 3-seconds duration providing an average fuel-flow rate of 12 cubic centimeters per second (0.015 lb/sec $\pm$10 percent).

In order to insure prompt and reliable ignition, the fuel was ignited with an electric-spark-type ignitor. The ignitor rod had a 1/4-inch diameter and was positioned as shown in figure 1. The spark gap between model surface and ignitor tip was located 1/4 inch upstream of the base

and was in line with an outboard fuel orifice. A repeating capacitance-type power supply (1-joule energy) provided a spark from rod to model surface 5 times a second. In previous experimental work the ignitor rod positioned as shown in figure 1 had only a small effect on the combustion and pressure fields.

Test Conditions and Procedure

The experiments were conducted at the Mach numbers of 2.47 and 2.96 with the wing at a 2° angle of attack. Before each combustion run the tunnel was brought up to the selected flow condition for the Mach number and stabilized. The total pressure was held as constant as practicable during the combustion. Average values of the free-stream parameters are summarized as follows:

Mach number	2.47	2.96
Total pressure, in. of Hg	47.53	67.92
Dynamic pressure, lb/sq ft	880	852
Nominal pressure altitude, ft	54,000	62,000
Total temperature, $^{\circ}$R	562	583
Reynolds number/ft	4.6×10^{6}	4.8×10^{6}
Dew point, $^{\circ}$F	<-35	<-50

RESULTS AND DISCUSSION

Description of Combustion and Associated Stream Disturbance

Steady, stable combustion of aluminum borohydride in the airstream adjacent to the lower surface of the wing was achieved under the test conditions. Nine combustion runs were made at Mach 2.47, and two were made at Mach 2.96. The combustion produced a very bright yellow-green color which was characteristic of the flames studied in the previous work (refs. 3, 5, and 6). A combination open-shutter and schlieren flash photograph of the flame and associated flow disturbances adjacent to the wing in the Mach 2.47 airstream is shown in figure 3. The flame is seen to be accompanied by an oblique shock system similar to that described in detail in reference 5 for combustion below a flat-plate model.

Because there has been limited experimental work to date on combustion in supersonic streams, little is known about the details of the combustion zone. The location of the heat-addition region below the wing is therefore not known with certainty. The schlieren and direct-photographic observations are not sufficient in this respect because although they show the geometry of the flame and the heated flow, they do not directly give information on the location or intensity of the combustion (or heat addition) region itself. The flame-shock structure and the

chordwise pressure-change profile caused by the heat addition (discussed in the next section) are most useful in defining the heat-addition region. They suggest that strong combustion took place in the upstream portion of the flame (within 1 or 2 in. of the fuel orifices) followed by a rapidly decreasing rate of combustion with distance downstream. In addition, visual observations of the flame far downstream of the base indicated that part of the fuel may have burned in the wake rather than below the wing. The schlieren photographs indicate that at both Mach numbers the shock waves from the leading edge of the wing and from the flame are reflected from the tunnel walls in such a manner that they cross the centerline of the tunnel downstream of the base in the heated-wake region of the wing. For the Mach 2.96 condition, the points of intersection of the shock waves with the wake are, of course, further downstream from the base than at Mach 2.47. For this reason only the Mach 2.96 base-pressure-change data are presented in the next section.

Static- and Base-Pressure Changes Caused by Combustion

The external combustion in the stream below the wing caused an increase in local static pressure along the lower surface and base. Average values (for all runs at each Mach number) of the local static-pressure increases for the two Mach number conditions are given in figures 4 and 5. Figure 4 shows the centerline chordwise pressure-change profiles along the lower surface of the wing for Mach 2.47 and 2.96. The chordwise distributions of static-pressure increase caused by the combustion are generally similar for the two Mach numbers. Only in the last inch before the base is there a significant difference in the shape of the two curves. The base pressures for the Mach 2.96 runs with and without combustion are shown in figure 5. The increase in base pressure caused by the combustion was about 100 percent.

The chordwise static-pressure profiles along the upper and lower surfaces of the wing with and without combustion are shown in figure 6. The Mach 2.47 data are shown in figure 6(a), and the Mach 2.96 data are shown in figure 6(b). The lower-surface static pressures during combustion were obtained by adding the average pressure changes caused by the combustion (fig. 4) to the corresponding static pressures without combustion. The upper-surface static pressures were not changed by the combustion. The measurements made at the spanwise station located 5 inches downstream of the fuel orifice (fig. 2) indicated that the static pressure

during combustion was approximately constant in the spanwise direction. Similar results were obtained with the flat-plate models of reference 5, which had identical fuel-orifice geometry.

The presence of reflected shock waves in the heated wake region of the wing requires that the pressure-change data be carefully inspected for any signs of wind-tunnel effects. With the exception of the small area just upstream of the base in the Mach 2.47 data of figure 4 (which will not be included in the lift calculations of the next section), the shape of the chordwise pressure-change distributions at both Mach numbers indicates that the static-pressure changes along the lower surface were not influenced by tunnel effects.

Lift and Drag Calculations

The lift and drag forces on the wings resulting from the surface- and base-pressure distribution can be calculated from the static-pressure data obtained with and without combustion in order to obtain an appraisal of the aerodynamic effects of external combustion. For simplicity, the lift and drag force components on the model will generally be presented directly in terms of pounds of force occurring on the model at the test conditions. Lift calculations were made for both Mach numbers. The drag calculations were made only for the Mach 2.96 runs.

For the calculations the pressure in the spanwise direction will be assumed constant at each chordwise station. As discussed previously, this is a reasonable assumption with the exception of a small area in the vicinity of the fuel orifices where the pressure field is, of course, more variable.

The lift and pressure-drag per foot span for the wing with and without external combustion were calculated from the data of figures 5 and 6. The nonburning friction drag was calculated from a skin-friction coefficient (based on plan form area) of 0.0035 obtained from unpublished wind-tunnel tests of this airfoil. During combustion the skin-friction coefficient was arbitrarily assumed to be the same as the nonburning value. The lift and drag calculations are summarized in the following table in which a positive force component indicates an upstream or thrust component, and a negative one indicates a downstream or drag component.

Lift and drag calculations	Mach number	Nonburning	Burning
Lift force, (lb/ft span)	2.47 2.96	60.7 50.3	117.7 113.0
Lift coefficient, C_L	2.47 2.96	0.06 .05	0.12 .12
Drag forces (lb/ft span) Pressure (upper) (lower) (base) Total Friction Total	 2.96	 1.4 -11.5 1.4 -8.7 -3.2 -11.9	 1.4 -15.7 2.8 -11.5 -3.2 -14.7
Lift-drag ratio, L/D	2.96	4.2	7.7

The lift calculations show that the lift of the wing was approximately doubled at both Mach numbers by burning aluminum borohydride injected into the airstream below the lower surface at an average rate of 0.015 pound per second. (This is equivalent to 372 Btu/sec at 100-percent combustion efficiency.) The Mach 2.96 calculations show that the pressure drag on the lower surface was actually increased during combustion, whereas the base drag decreased. The increase in pressure drag on the lower surface resulted from starting the addition of heat far upstream (approximately 0.2 chord) in order to maximize the lift component. Because of the airfoil shape most of the pressure increase was adjacent to the portion of the surface with a forward-facing component. However, because of the large lift increases and the increase in base pressure, the effective lift-drag ratio was still nearly doubled.

Because the chordwise pressure profiles during combustion appear to be reasonably free of wind-tunnel effects, the calculated values of lift with external combustion should represent values that could be achieved in free flight. Values of the effective lift coefficient C_L during combustion at each Mach number are therefore included in the above table. It should be pointed out also that aerodynamic parameters such as L/D and C_L do not have their usual meaning. In addition to the fact that their values depend on the addition of energy to the stream, they are strictly speaking point values for a specific flight condition and heat addition. This is because the combustion and flow fields are interdependent. For example, doubling the dynamic pressure may not necessarily double the lift (at the same Mach number and fuel input), because the combustion will also be affected. However, the values of

these parameters are presented here in order to provide a familiar basis of comparison at the test conditions.

Further, although the increase in lift can be sizeable, a comparison with the conventional wing plus internal burning ramjet engine indicates that, at the low supersonic Mach numbers of this test, external burning requires a considerably higher fuel consumption rate to obtain a given lift increment.

Comparison with Theory

In order to compare the experimental results of this investigation with those predicted by theoretical calculations, the geometry and intensity of the heat-addition region below the wing must be defined. In the analytical calculations of reference 2 this is simply a matter of arbitrary definition. However, in the experimental work of this report, a definition of the heat-addition region is difficult, because the locus of the actual heat addition in the flame is not known.

A useful approach to the problem of overcoming this difficulty was obtained from the theoretical analyses of references 8 and 9. In reference 9 a comparison is made of lift coefficients obtained by linearized solutions for heat addition below a circular-arc airfoil with those obtained by more exact graphical computations in reference 2. The method used in reference 9 to compare the two types of analytical solutions is particularly adaptable for comparison of the experimental data with the analytical results. This is because the method circumvents the necessity of knowing the exact locus of the heat addition and uses instead the resultant flow deflection as a measure of heat addition.

This approach is used in figure 7 to compare theoretical and experimental values of lift obtained for heat addition below a supersonic airfoil. The lift parameter, product of the lift coefficient and $\sqrt{M^2 - 1}$ (M is the Mach number), is the ordinate, the flow-deflection parameter is the abscissa. The flow-deflection parameter is a quantity related to the deflection of the supersonic stream by the airfoil angle of attack and the heat-addition region. The points plotted in figure 7 with circular symbols were obtained from the analytical results of reference 2 for a 5-percent thick circular-arc airfoil. The experimental values for the 6-percent thick airfoil of this report at Mach numbers of 2.47 and 2.96 are plotted with solid square symbols. The results of the linearized theory of reference 9 are given by the straight line. Figure 7 shows that there is good agreement between the experimental values of lift and the analytical lift values of reference 2. The experimental data points and analytical calculations of reference 2 follow the same trend in deviating from the linearized results at large values of the flow-deflection parameter (i.e., large heat additions).

All parameters used in plotting the experimental lift data on figure 7 were obtained from measured quantities. The lift coefficients for each Mach number have already been given in the preceding section. In the

flow-deflection parameter $\left(\alpha + \frac{1}{2} f\delta_h\right)$ the angle of attack α was fixed at 2°. The fraction of chord f below which the heat is added was determined by the fraction of chord over which a static-pressure rise was measured. Values of $f = 0.6$ at Mach 2.47 (small pressure rise near base neglected) and $f = 0.8$ at Mach 2.96 were obtained from figure 6. The angle through which the flow was deviated by the heat addition δ_h was determined from the angle of the oblique shock-wave system generated by the flame and by the Mach number just ahead of the flame shock wave. The angle of the flame shock system was determined from schlieren photographs (such as fig. 3). Within the accuracy of measurement, δ_h was found to be about 10° at both Mach numbers.

Basic to the assumptions used in this type of analysis is the concept of pressure waves produced by the expansion of the volume occupied by the the heated gas. This results in a velocity component in the external supersonic stream which is normal to the airfoil surface. Thus, the heat-addition region deflects the external flow as if it were effectively a wedge in the stream. Experimental evidence of this type of deflection of the stream was presented in reference 5 for heat addition below a flat-plate model at zero angle of attack. In fact, the effective wedge angle of the heat-addition region below the flat plate was also 10° for a comparable fuel-injection rate. For the airfoil at angle of attack, the heat-addition increases the effective inclination of the lower surface. A theoretical relation for the effective increase in surface inclination δ_h is derived in references 8 and 9 in the form

$$\delta_h = \frac{\gamma - 1}{\gamma} \frac{q}{p_o V_o} \tag{1}$$

where

q rate of heat addition per unit area
p_o free-stream static pressure
V_o free-stream velocity
γ ratio of specific heats

Since δ_h was a measured quantity in this experiment along with p_o and V_o, the theoretical rate of heat addition per unit area q could be calculated from equation (1). The average value of q calculated from equation (1) from the Mach 2.47 data is 327 Btu per second per square foot. This quantity can be compared with the maximum (100-percent combustion efficiency) energy available from the fuel injected below the wing. The maximum energy available from the fuel injection (0.015 lb/sec $\times$24,800 Btu/lb) was 372 Btu per second which is equivalent to 572 Btu per second per square foot (for $f = 0.6$). In consideration of the over-all accuracy of the calculations, the primary conclusion which can be drawn from this comparison is that the magnitudes of the theoretical and experimental values of q are comparable.

CONCLUDING REMARKS

The results of this investigation show that significant improvement in the lift-drag ratio can be achieved by moderate heat addition below a wing. Because heat was added below a large fraction of the airfoil, the emphasis was primarily on increasing the lift. The experimental values of lift were in good agreement with those predicted by analytical calculations for a comparable heat addition. An alternate approach would have been to start the heat-addition region at a point closer to the base of the airfoil. Then the associated pressure rise would have been primarily adjacent to rearward facing surfaces, and the drag-reduction (or thrust) component of the force vector would have beem emphasized.

The results of this report indicate the aerodynamic effects of adding heat below a conventional wing which has good aerodynamic qualities initially. Because the aerodynamic effects of external heat addition are a function of the shape of the wing (or other aerodynamic body), an alternate approach would have been to select on the basis of theory a shape for the experimental investigation which would appear to be optimum for a particular mission. For example, it should be possible to select for experimental testing a combination of heat addition and airfoil geometry designed to give zero net drag (i.e., thrust minus drag = 0) in combination with usable lift during cruise conditions.

An investigation was recently completed at the NASA Lewis Research Center by R. G. Dorsch, Harrison Allen, Jr., and Murray Dryer in which a small (25-in. chord) flat-plate external-combustion model was tested in a very large (10 by 10 ft) supersonic tunnel in order to minimize the possibility of tunnel-wall effects. Analysis of the large-tunnel data showed that there was good agreement with the data reported herein. It is considered, therefore, that the small-tunnel external-combustion data of this report are substantially free of tunnel effects.

A comparison with internal burning ramjet engines indicates that external burning requires considerably higher total fuel consumption rates at the low supersonic Mach numbers of this test. However, theoretical considerations indicate a steadily decreasing over-all efficiency for the conventional ducted ramjet engine (with subsonic combustion) at Mach numbers between 5 and 10 whereas the over-all efficiency of external burning remains relatively constant in this speed range. It therefore appears that external burning will be competitive at higher Mach numbers.

It is recognized that an internal combustion ramjet engine using supersonic combustion may combine some of the advantages of both types of engines. This engine should therefore also be considered as a future possibility when assessing the relative merits of various air-breathing propulsion schemes for use at hypersonic Mach numbers.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, April 11, 1958

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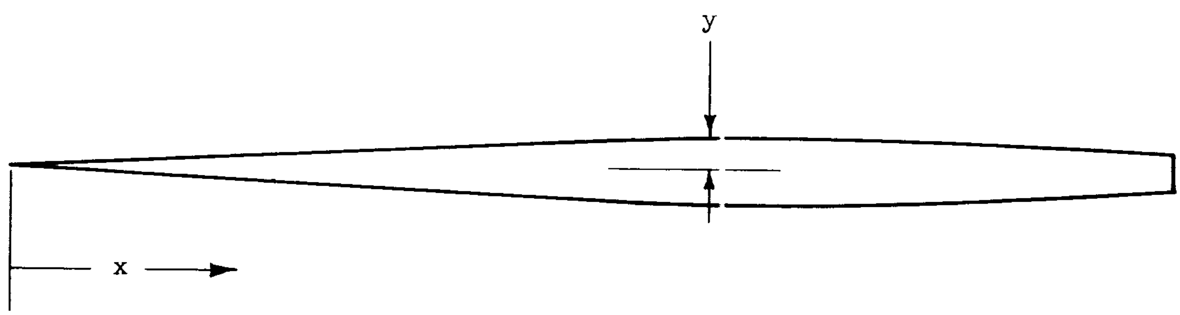
TABLE I. - COORDINATES OF AIRFOIL SECTION



x, in.	y, in.
0	0
.650	.040
1.300	.079
1.950	.119
2.600	.159
3.250	.196
3.900	.232
4.550	.267
5.200	.298
5.850	.326
6.500	.350
7.150	.370
7.800	.382
8.450	.389
8.723	.390
9.100	.389
9.750	.378
10.400	.361
11.050	.339
11.700	.310
12.350	.275
13.000	.231

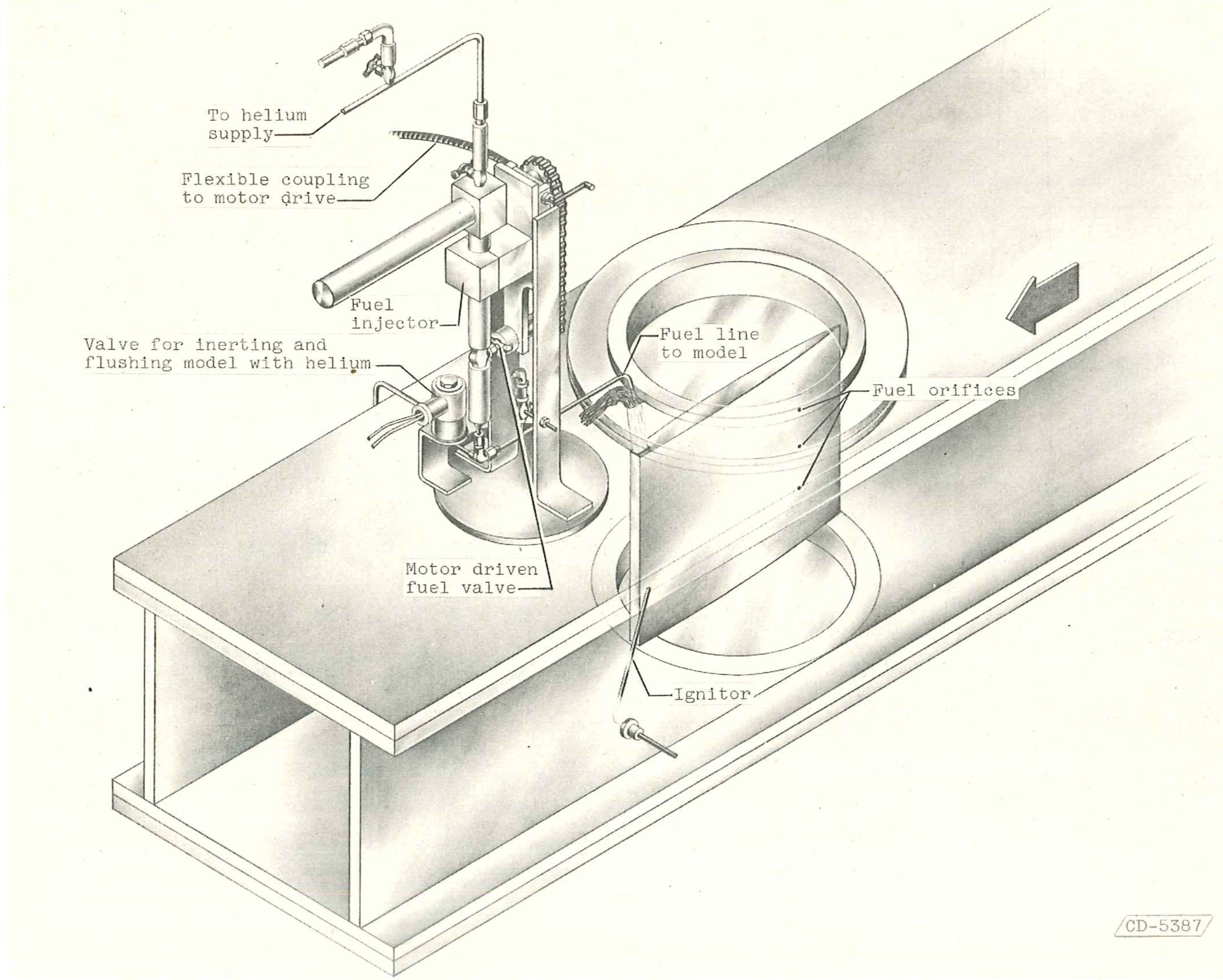


Figure 1. - Sketch of wing mounted in wind tunnel, fuel-injection system, and ignitor-rod location.

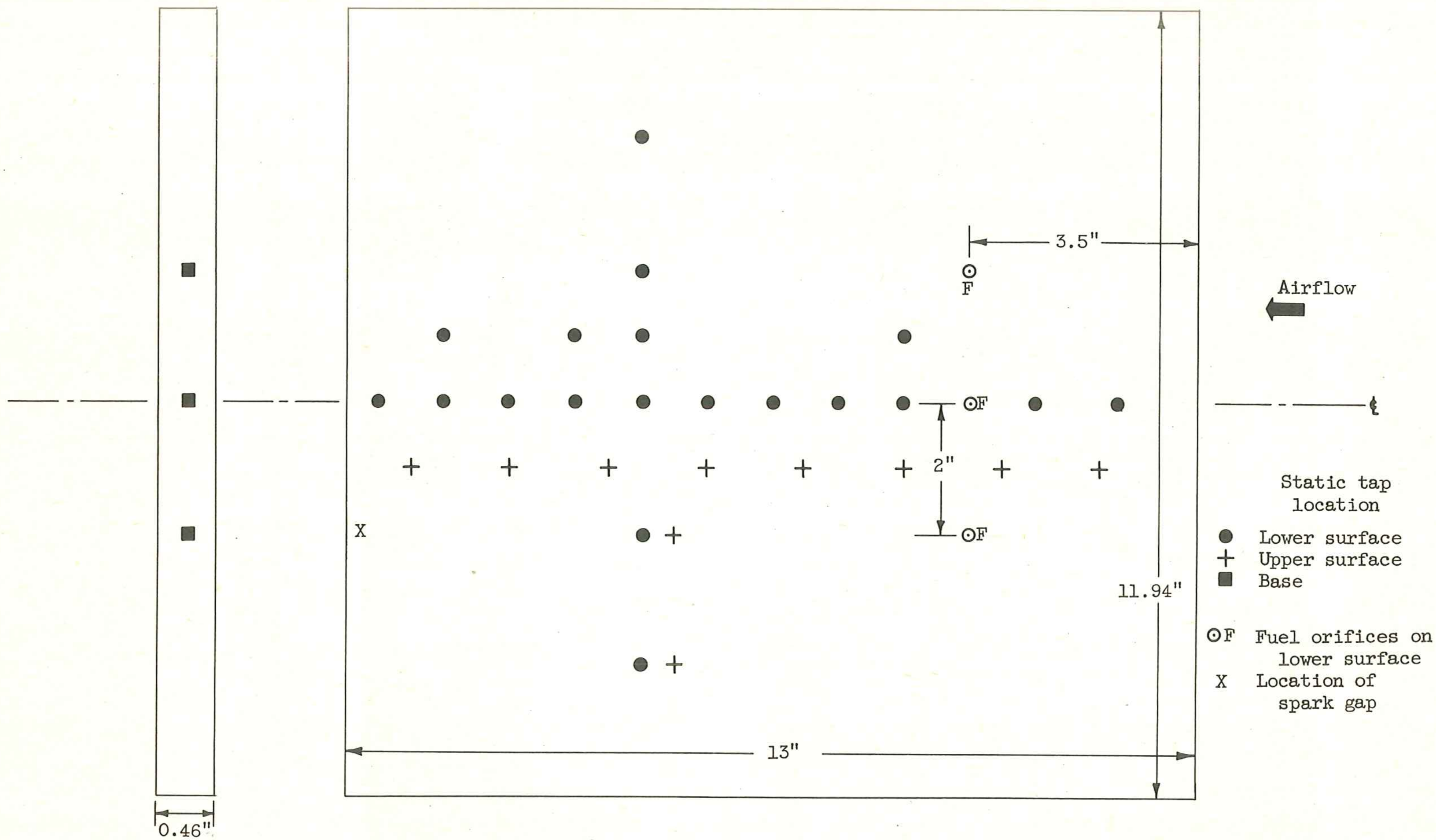


Figure 2. - Arrangement of static-pressure taps and fuel orifices on wing surface and base.

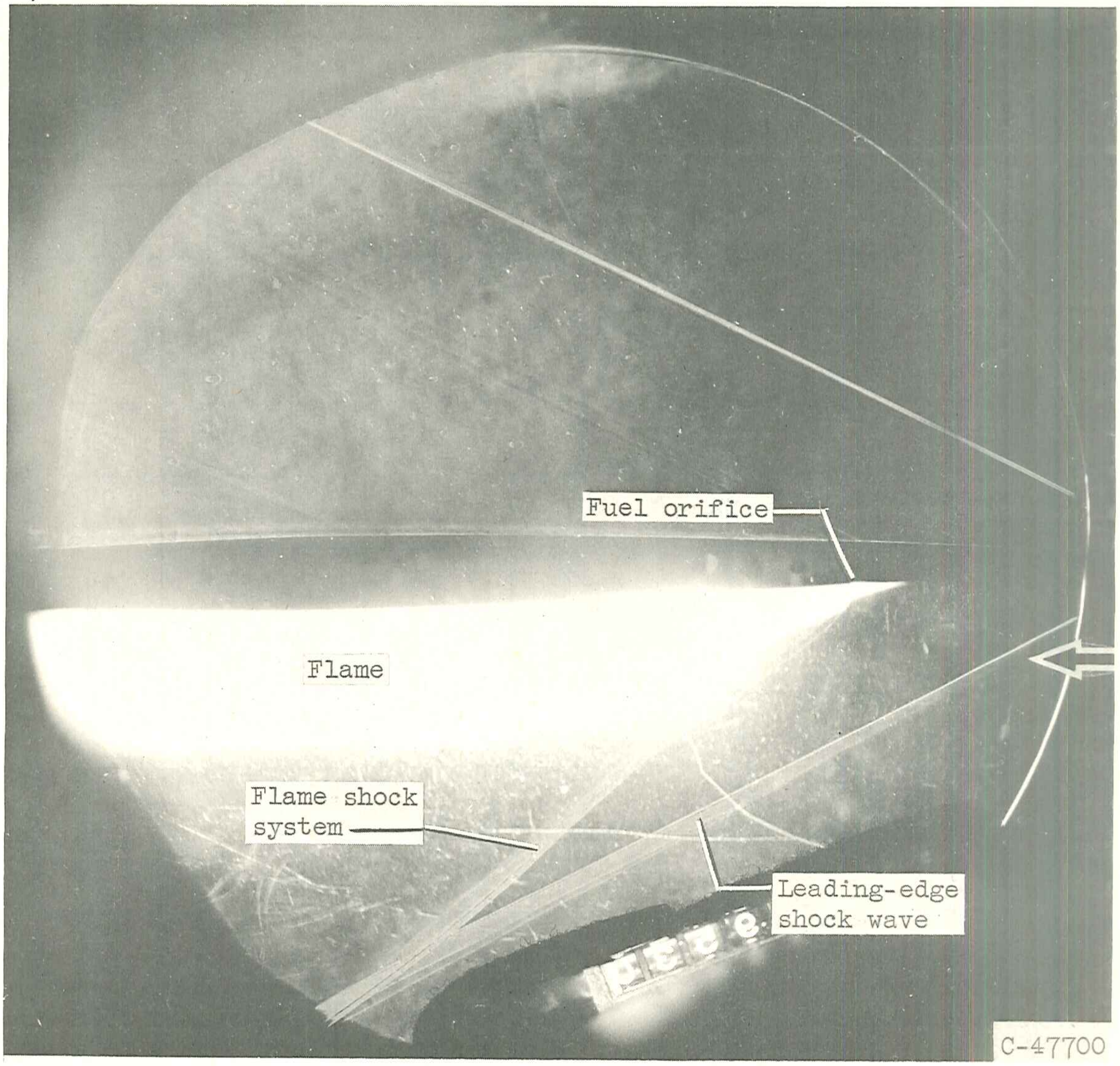


Figure 3. - Combination open-shutter and schlieren flash photograph of flame and associated shock-wave system in stream below surface of wing. Mach number, 2.47; angle of attack, 2°.

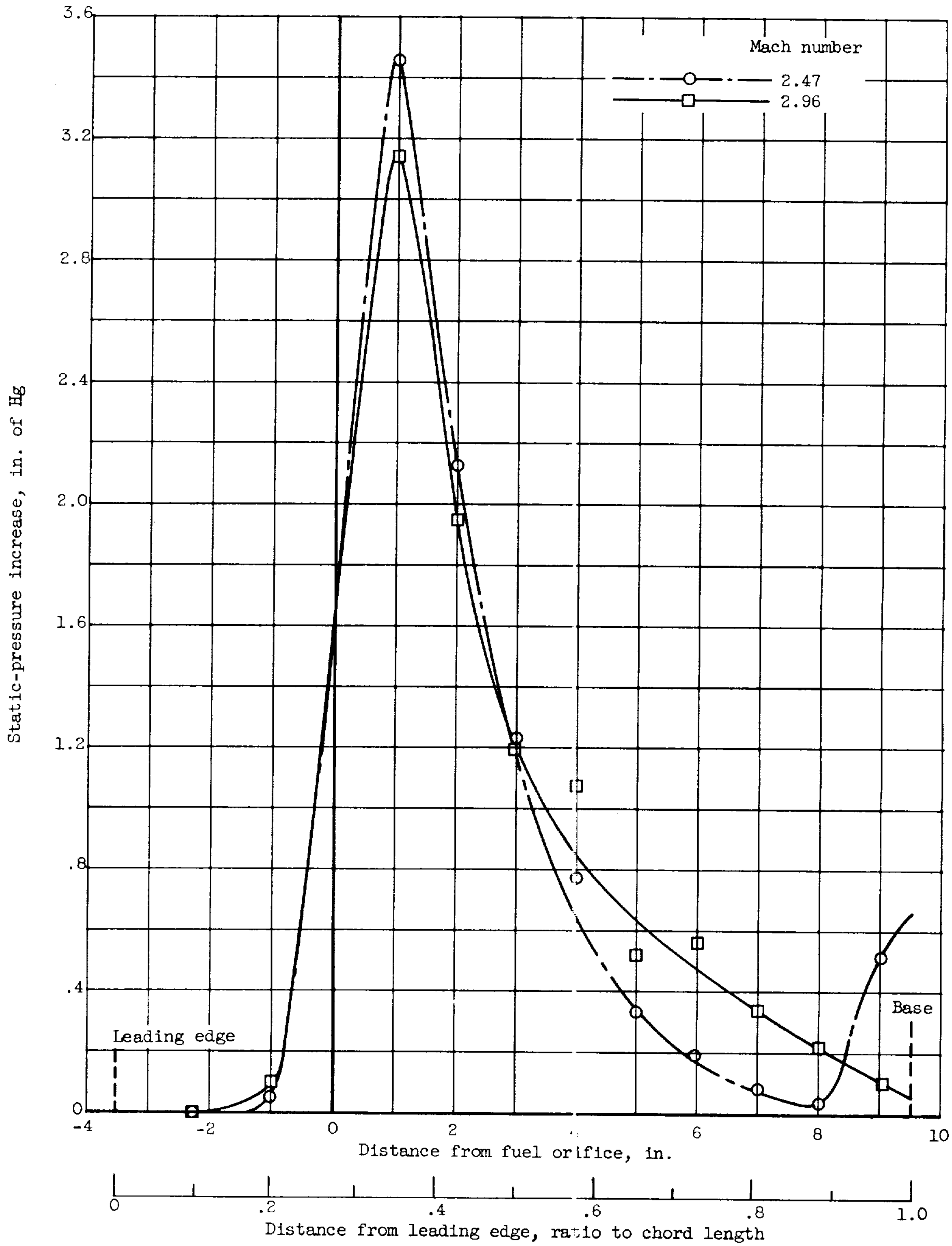


Figure 4. - Average chordwise static-pressure increases along lower surface caused by combustion below wing at a 2° angle of attack.

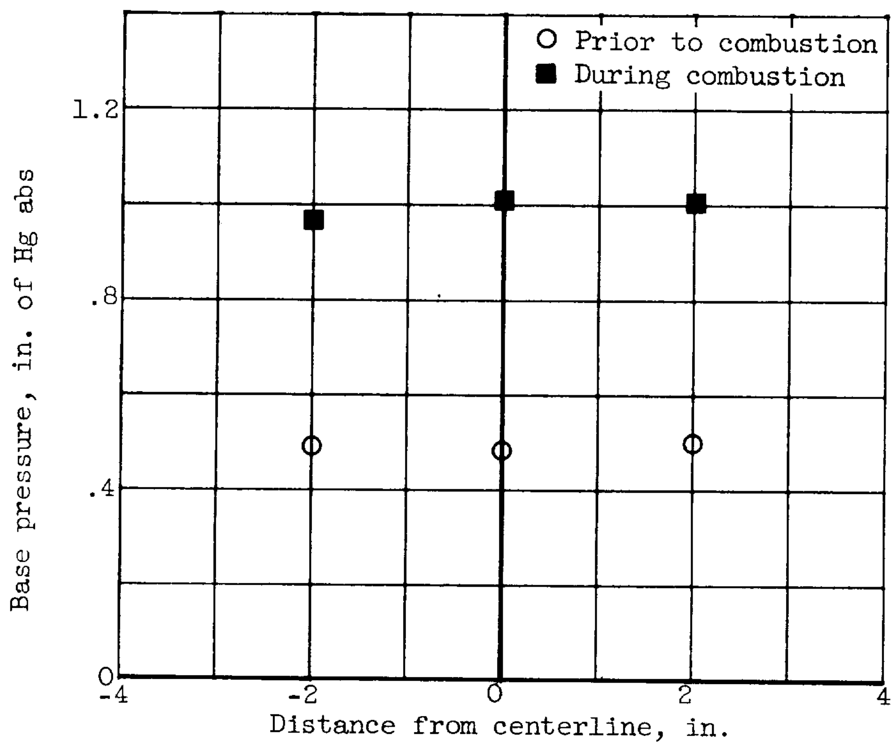


Figure 5. - Comparison of base pressures for Mach 2.96 runs with and without external combustion below wing in terms of absolute pressures.

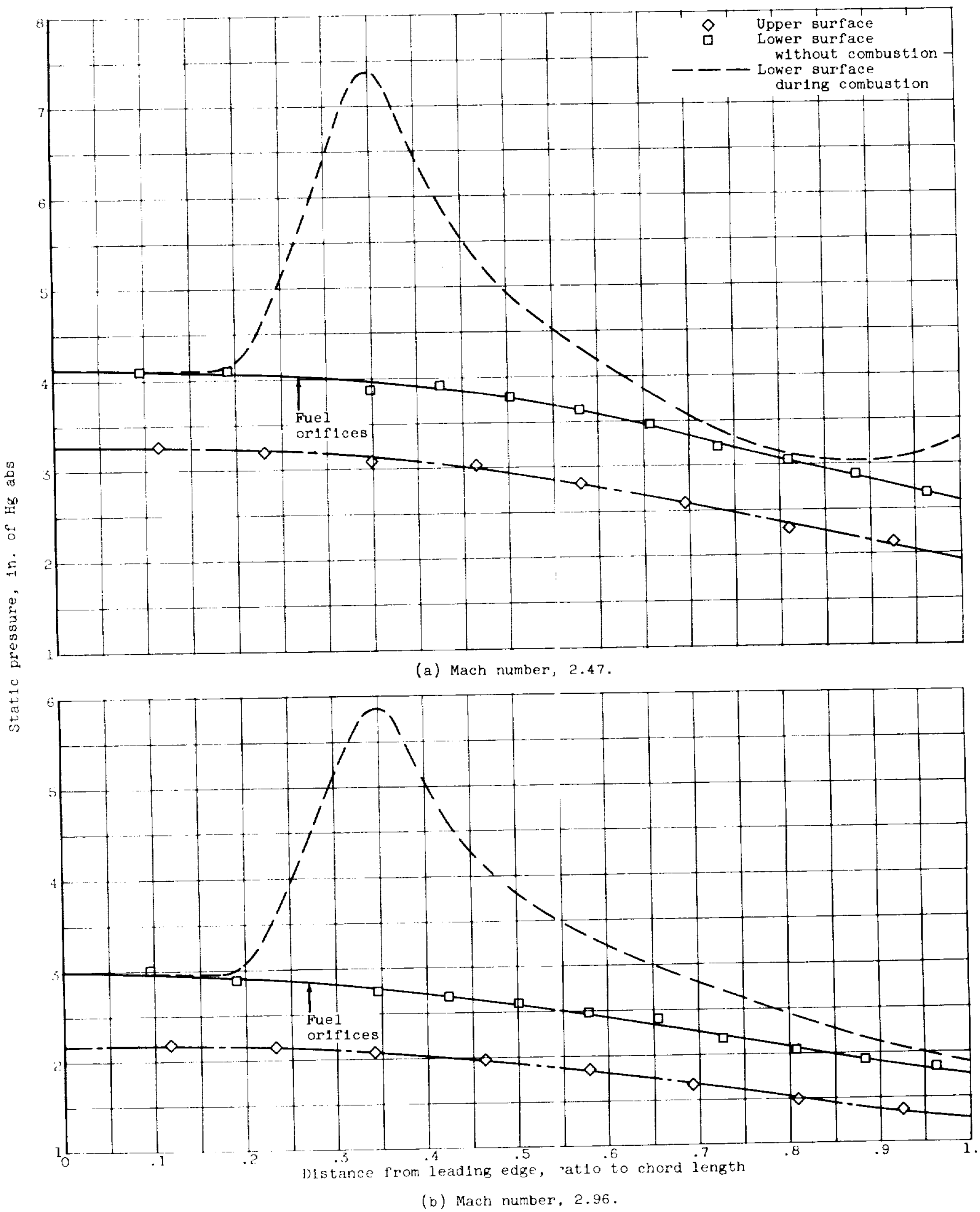


Figure 6. - Chordwise static pressures with and without external combustion. Angle of attack, 2°.

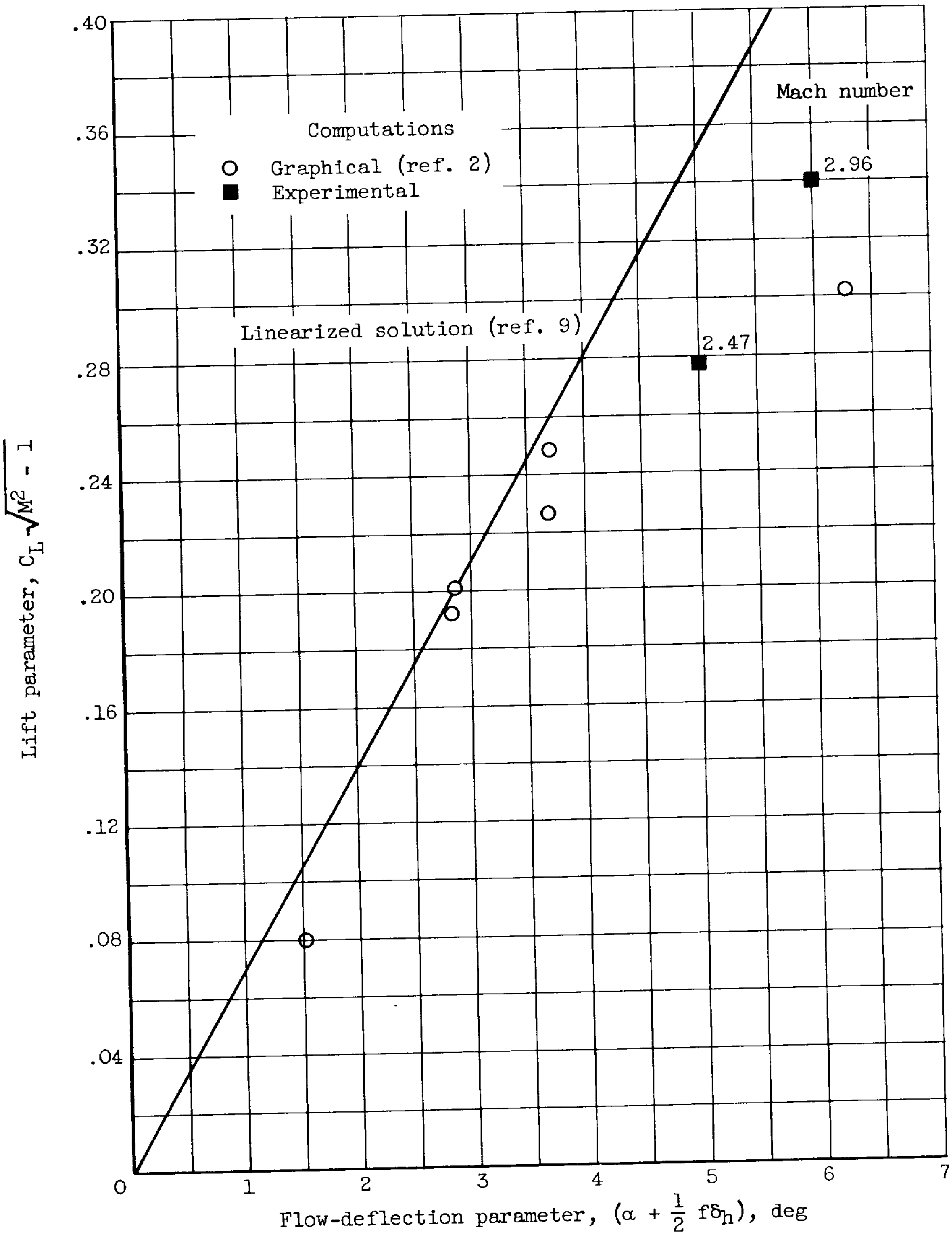


Figure 7. - Comparison of experimental and theoretical values of lift of supersonic airfoil with heat addition adjacent to lower surface.